Europa Clipper Mission: Preliminary Design Report

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Abstract—**Europa, the fourth largest moon of Jupiter, is believed to be one of the best places in the solar system to look for extant life beyond Earth. Exploring Europa to investigate its habitability is the goal of the Europa Clipper mission.**

The Europa Clipper mission envisions sending a flight system, consisting of a spacecraft equipped with a payload of NASA-selected scientific instruments, to execute numerous flybys of Europa while in Jupiter orbit. A key challenge is that the flight system must survive and operate in the intense Jovian radiation environment, which is especially harsh at Europa.

The spacecraft is planned for launch no earlier than June 2023, from Kennedy Space Center, Florida, USA, on a NASA supplied launch vehicle. The mission is being implemented by a joint Jet Propulsion Laboratory (JPL) and Applied Physics Laboratory (APL) Project team. The project recently held its Project Preliminary Design Review and in early 2019 NASA will consider approving the mission for entry into Phase C, the Detailed Design phase. A down-selection to one launch vehicle by NASA is anticipated sometime before Project Critical Design Review.

This paper will describe the progress of the Europa Clipper Mission since January 2018, including maturation of the spacecraft, subsystem and instrument preliminary designs, issues and trades, and planning for the Verification & Validation phase.

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1. INTRODUCTION

Europa's subsurface ocean is a particularly intriguing target for scientific exploration and the hunt for life beyond Earth. The 2011 Planetary Decadal Survey, *Vision and Voyages*, states: "Because of this ocean's potential suitability for life, Europa is one of the most important targets in all of planetary science" [1]. Investigation of Europa's habitability is intimately tied to understanding the three "ingredients" for life: liquid water, chemistry, and energy. The Europa Clipper mission would investigate these ingredients by comprehensively exploring Europa's ice shell and liquid ocean interface, surface geology and surface composition to glean insight into the inner workings of this fascinating moon. A mission to land directly on Europa's surface would be a scientifically desirable future step, but current data regarding the Jovian radiation environment, potential landing site hazards, and potential safe landing zone locations are insufficient. Therefore, an additional goal of the Clipper mission would be to characterize the radiation environment near Europa and investigate scientifically compelling sites for hazards, to inform a potential future lander mission.

To achieve these habitability assessment and reconnaissance goals, the Clipper mission envisions sending a spacecraft, equipped with a payload of NASA-selected scientific instruments, to execute numerous flybys of Europa while in Jupiter orbit. A key challenge is that the spacecraft and instruments (the "flight system") must survive and operate in the harsh Jovian radiation environment, which is especially intense at Europa. Therefore, the innovative design of this multiple-flyby science tour is an enabling feature of this mission: by minimizing the time spent in the radiation environment the spacecraft complexity and cost has been significantly reduced compared to previous mission concepts.

Europa Clipper is planned for launch from Kennedy Space Center in Cape Canaveral, Florida, USA, no earlier than June, 2023. Clipper will launch on a NASA supplied launch

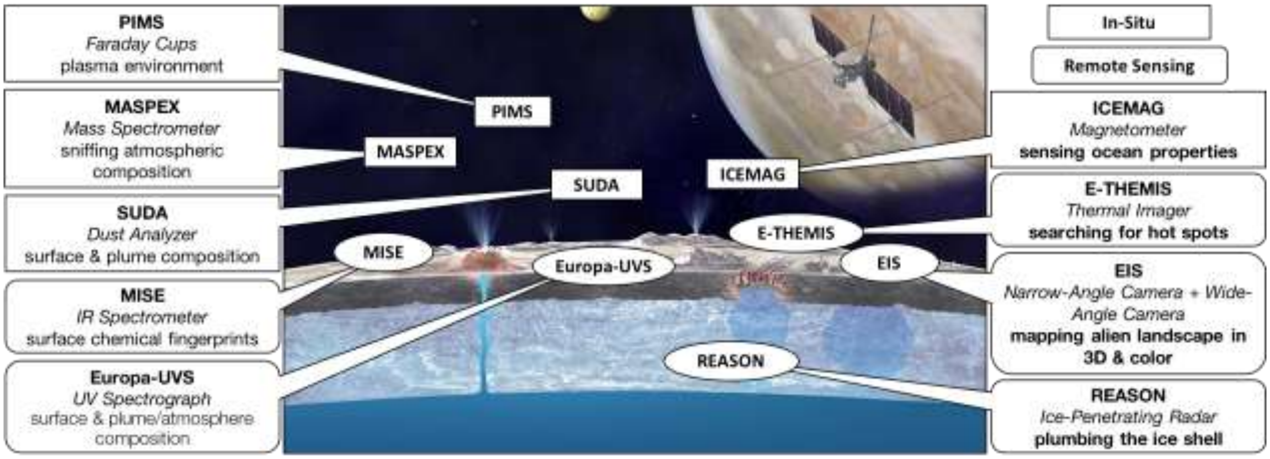

Figure 1: Clipper Instruments and Science Investigations

vehicle, and a down-selection to one launch vehicle by NASA is anticipated sometime before Critical Design Review (CDR). Selection of the NASA Space Launch System (SLS) would enable a direct-to-Jupiter trajectory, which would remove the need to fly through the inner solar system on a gravity assist trajectory to Jupiter, significantly shortening the cruise phase.

In August 2018, the Clipper mission held its Project Preliminary Design Review (PDR), capping a year of approximately two dozen PDRs at the Flight System, Subsystem, Instrument, and Mission System levels. Challenges with the in-house development of the solar array led the project to decide to procure the array instead. This change, in addition to accommodation of the radar antennas, caused the solar array preliminary design to be delayed. These challenges resulted in an incomplete Project PDR and a follow-up review is planned for early 2019.

This paper is the fourth to describe progress from pre-formulation through the present. See [2], [3], and [4] for historical context.

In November 2018 NASA Headquarters directed the project to prepare for a launch no earlier than 2023 – a change from the original direction to prepare for an earliest launch date of 2022. Because the project is conducting planning exercises for the new date as of December 2018, the remainder of this paper will reflect the previous 2022 launch date.

2. Science and Instrument Updates

The science objectives driving the Clipper mission remain unchanged; this section will review those objectives, Clipper's instrument suite, and provide highlights of Instrument progress and design updates in the last year.

Science Objectives

The science objectives are the principal drivers of the Europa Clipper mission and remain fully consistent with planetary science objectives defined by NASA. The science objectives, in order of priority, are:

1. Ice Shell and Ocean: Characterize the ice shell and any subsurface water, including their heterogeneity, ocean properties and the nature of surface-ice-ocean exchange. Map the vertical subsurface structure beneath ≥50 globally distributed landforms to ≥3-km depth *to understand the distribution of subsurface water and processes of surface-ice-ocean ex- change.* Constrain the average thickness of the ice shell, and the average thickness and salinity of the ocean, each to ±50%.

2. Composition: Understand the habitability of Europa's ocean through composition and chemistry. Create a compositional map at ≤10-km spatial scale, covering ≥70% of the surface *to identify the composition and distribution of surface materials.* Characterize the composition of ≥50 globally distributed landforms, at ≤300-m spatial scale *to identify non-ice surface constituents including any carbon-containing compounds.*

3. Geology: Understand the formation of surface features, including sites of recent or current activity and characterize high-science-interest localities. Produce a controlled photomosaic map of ≥80% of the surface at ≤100-m spatial scale *to map the global distribution and relationships of geologic landforms.* Characterize the surface at ≤25 m spatial scale, and measure topography at ≤15-m vertical precision, across ≥50 globally distributed landforms *to identify their morphology and diversity.* Characterize the surface at ∼1-m scale to determine surface properties, for ≥40 sites each ≥2 km x 4 km.

4. Recent Activity: Search for and characterize any current activity, notably plumes and thermal anomalies, in regions that are globally distributed.

Science Payload Overview and Updates

During the past year, the payload team updated their requirements, including adding calibration requirements and

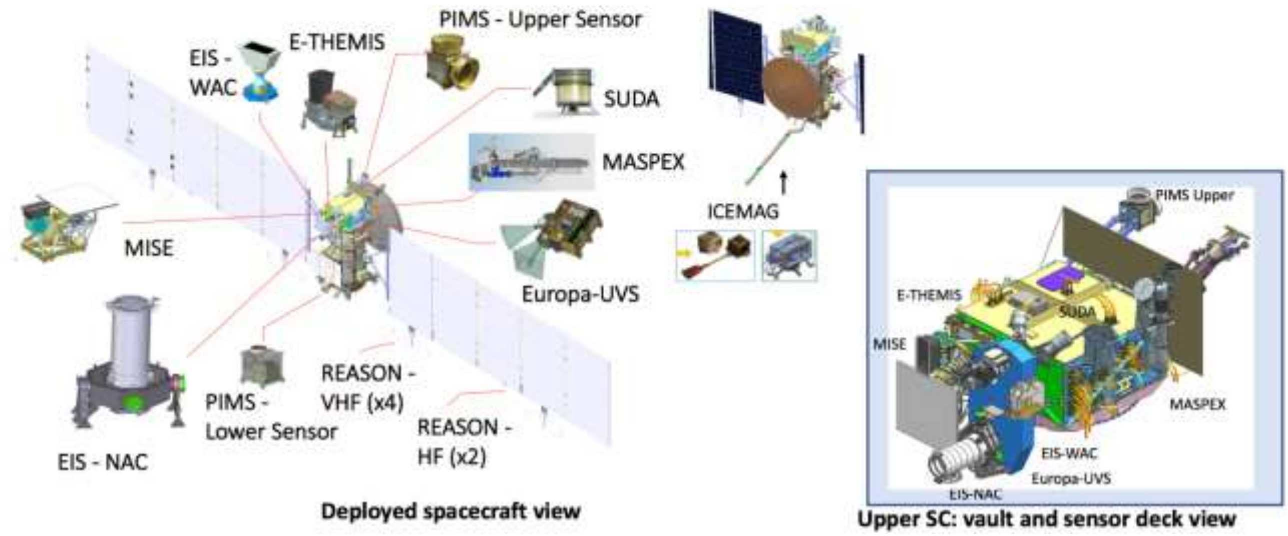

Figure 2: Clipper Instruments on the Spacecraft

release of baselined level 4 instrument requirements. Mass and energy allocations were updated and data allocations were added. The instruments all passed their PDRs and the teams are now working on closure of assigned PDR action items and preparation for Instrument CDRs.

REASON (Radar for Europa Assessment and Sounding: Ocean to Near- Surface)—this radar instrument is mounted on Clipper's solar arrays. It will use shallow and deep sounding to characterize the structure of Europa's ice shell and surface features. Updates: After qualification testing of materials, REASON removed their heaters from the matching networks at the antennas. Additionally, REASON worked with the solar array vendor to determine how conductivity and grounding requirements will be met, and finally, the team removed dielectric from Very High Frequency (VHF) antenna design to avoid electrostatic discharge concerns.

EIS (Europa Imaging System)—really two instruments, EIS consists of a Near-Angle Camera (NAC) and a Wide-Angle camera (WAC). Together, these instruments provide visible maps of Europa's surface and geology, and hunt for scientifically-compelling landing sites. Updates: EIS worked to revise the gimbal system for the NAC camera to adjust for higher loads and increased reliability and revised to a single launch lock design. Additionally, EIS removed the cross-strapping between the NAC and WAC cameras to reduce complexity and save mass. Lastly, EIS moved the cover position to remove incursion into MISE's stray light field-of-view.

Europa-UVS (Europa Ultraviolet Spectrograph)—this instrument characterizes plumes erupting from Europa's surface and also investigates composition and chemistry of Europa's atmosphere. No substantial updates to the UVS design were made in the last year, due to the high maturity of the design, driven by commonality with the UVS instrument on the European Space Agency's JUpiter ICy moons Explorer (JUICE) mission.

SUDA (SUrface Dust Mass Analyzer)—collects small particles from Europa's atmosphere during flybys and, by analyzing these particles, can map Europa's surface composition. Updates: SUDA incorporated the mounting bracket into their interface design and added a detector buffer amplifier to improve their signal-to-noise ratio.

MISE (Mapping Imaging Spectrometer for Europa)—this instrument acquires infrared image data of Europa's surface, which allows analysis for the presence of organic compounds and acid hydrates, salts, and other materials relevant to the habitability of Europa's ocean. Updates: MISE changed to a single cryocooler design, which improved resources, but caused a major change to the instrument's configuration and mechanical interface with the spacecraft. MISE also switched to a CaF2 detector technology to better survive the radiation environment.

E-THEMIS (Europa Thermal Emission Imaging System)—provides thermal imaging of the Europan surface to hunt for thermal anomalies and active plumes. Updates: In the last year, E-THEMIS has selected the microbolometer vendor and updated the design of their radiator.

MASPEX (MAss Spectrometer for Planetary EXploration / Europa)—this neutral mass spectrometer will determine chemical composition of Europa's atmosphere and exosphere, focusing on major volatiles and key organic compounds. Updates: MASPEX has relocated their cryocooler, radiator assembly, and reflectron pulser. MASPEX has also incorporated the mounting bracket into the MASPEX mechanical design.

ICEMAG (Interior Characterization of Europa using Magnetometry)—a magnetometer consisting of four sensors spread along a 5-m boom, ICEMAG will measure magnetic fields near Europa, which provides insight into ocean and ice shell properties and ocean conductivity. Updates: This year, ICEMAG separated the cable along the boom into multiple cables to reduce torque at the boom joint and reduced required heater power after successful completion of qualification tests of fiber optic cable.

PIMS (Plasma Instrument for Magnetic Sounding)—PIMS measures density, flow, and energy of ions and electrons encountered by the spacecraft as it travels around Jupiter and Europa. PIMS provides insight into ice shell thickness, ocean depth, and salinity. Updates: PIMS removed their deployable cover after analysis shows it was not necessary. PIMS also moved their electronics out of the vault and into the sensor head.

The Clipper flight system design also supports a gravity science investigation which will be used to confirm the presence or absence of a global subsurface ocean beneath Europa's icy crust. Gravity field measurements will use the Clipper Telecom System's two-way Doppler measurements during each flyby, combined with altimetry measurements from REASON, to determine the Europa tidal Love number k_2 with a accuracy of less than 0.06.

3. MISSION DESIGN UPDATES

This section will discuss updates to the mission trajectory and updates to the mission's major scenarios and phases.

Launch Vehicle Compatibility

Europa Clipper continues to maintain a design that is compatible with launch onboard either the SLS or a non-SLS heavy-lift expendable launch vehicle (ELV) in 2022, with a backup launch opportunity in 2023. Clipper's Launch Vehicle options include the SLS Block 1, the Delta IV Heavy and the Falcon Heavy. The SLS Block 1B was removed from consideration following a May 2018 NASA Headquarters memo which stated that a potential Clipper SLS launch would use the Block 1 variant only [5].

Interplanetary Trajectories

Due to the very different performance capabilities of launch vehicles under consideration, very different interplanetary trajectories are required to reach Jupiter with sufficient mass to interrogate Europa's habitability. The SLS Block 1 would afford the capability to travel to Jupiter on a direct trajectory in 2.5-2.7 years, while the largest non-SLS ELVs would require the utilization of an EVEEGA (Earth-Venus-Earth, Earth Gravity Assist) trajectory in 2022 that would extend the interplanetary cruise duration to 7.6 years, as shown in Figure 3 and Figure 4. In addition, for the direct trajectory case, a dual-arrival date strategy would be implemented in order to maximize delivered mass to the Jupiter system.

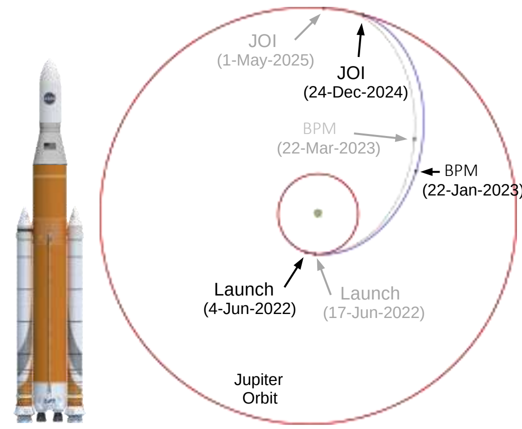

Figure 3: Earth-Jupiter direct trajectory utilizing the SLS Block 1

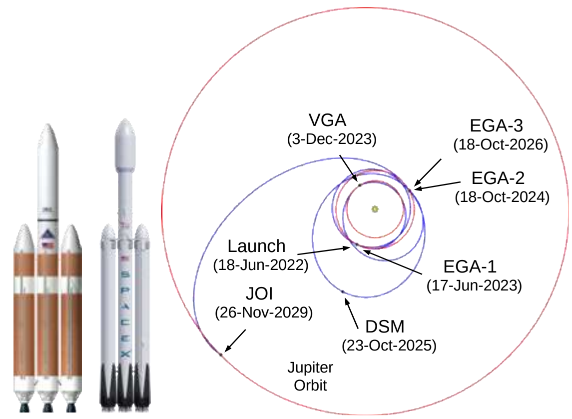

Figure 4: EVEEGA trajectory utilizing a non-SLS ELV (Delta IV Heavy or Falcon Heavy)

Tour Design Update

The Europa Clipper mission is predicated on the developed capability to efficiently obtain global-regional coverage of Europa (i.e., data sets at the regional scale, distributed across Europa globally) via a complex network of Europa flybys while in Jupiter orbit [6][7][8][9]. These orbits are highly elliptical, designed to minimize the time the Flight System spends in the region of intense radiation Europa is continually immersed in. The key mission design strategy is to dip in just low enough to skirt Europa's orbit to collect a high volume of quality Europa data and then quickly escape the most intense portions of the radiation environment, thus enabling the vast majority of the orbit to downlink high volumes of Europa data without significant radiation exposure. In addition, the time outside the harsh radiation environment provides time to react to potential anomalies and discoveries.

The current reference science tour is still 17F12_V2.[1] This tour has been designed to maximize science return to meet Level-1 science requirements by way of well over 300 science measurement and calibration requirements levied on the mission design (i.e., trajectory design, mission planning

[1]Tour names follow a specific convention in order to facilitate management of the different options, analyses, and evolution of the mission.

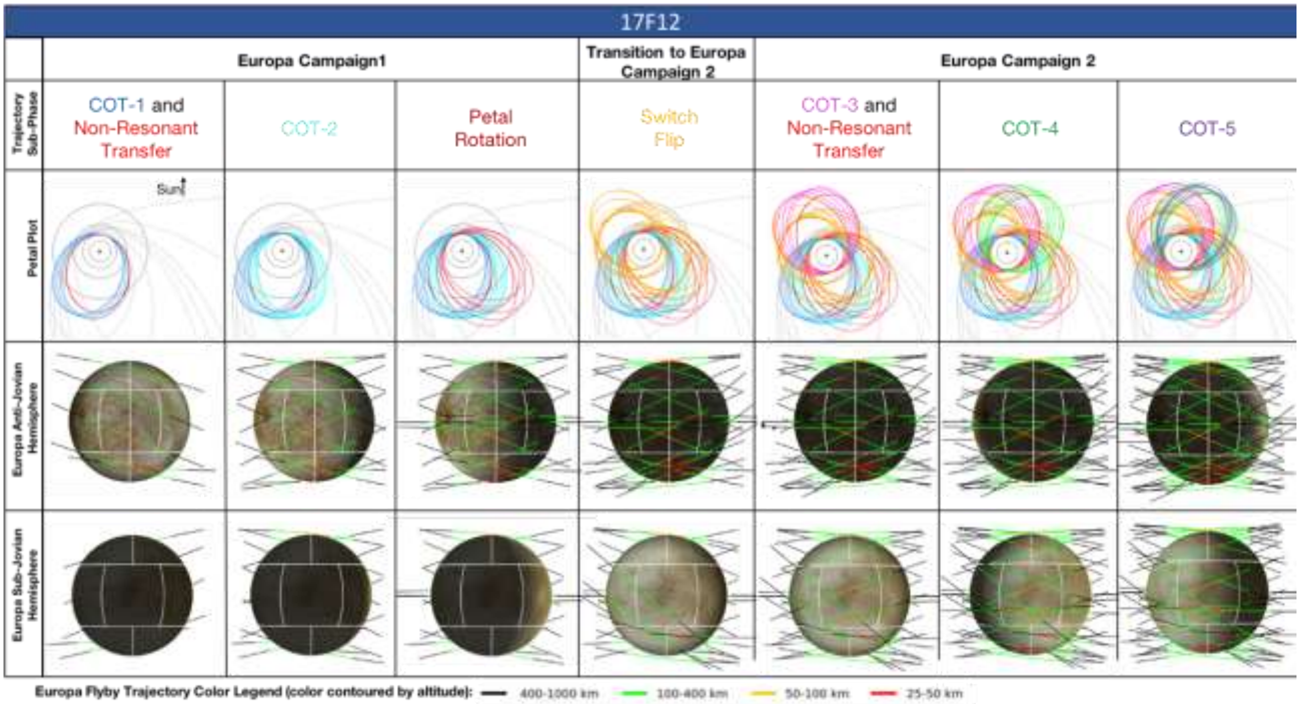

Figure 5: Science Tour Sub-Phases

and navigation design) as well as over 100 requirements stemming from project policies, planetary protection, and the evolved capability and characteristics of the flight system and mission operations system. The 17F12_V2 trajectory utilizes a 2022 Earth-Jupiter direct interplanetary trajectory and consists of 46 Europa, 4 Ganymede, and 9 Callisto flybys over the course of 3.7 years. It has a total ionizing dose (TID) of 2.5 Mrad[3]. The average period of each Jovian orbit is 20 days, and the typical time between Europa science flybys is 14.2 days. The 17F12_V2 trajectory was the baseline for all subsystem and project PDRs. The science tour sub-phases can be seen graphically in Figure 5.

Mission Scenarios and Phases

The high-level strategies conceived to successfully carry out the mission have remained stable in the last year, with only small refinements taking place as the Flight System and payload design continue to mature. This section will provide an overview of the major events and phases in the mission and significant updates and challenges associated with them.

Launch—Launch is one of two critical events during the mission, and completes when the Spacecraft has achieved a thermally safe, power-positive, and communications-enabled state following Launch Vehicle (LV) separation. This requires a significant amount of autonomy and robustness in the spacecraft design and on-board behaviors, as the spacecraft must achieve this state without ground in the loop. The Telecom subsystem will be configurated for uplink and downlink following LV separation, but autonomous behaviors ensure that the launch day critical events complete in the event that ground contact cannot be immediately established.

These behaviors are built directly into flight software, and incorporate all of the stages necessary to arrive at a thermally safe, power-positive, and comms-enabled state. These stages include: monitor breakwires to detect LV separation; configure the communications and navigation hardware for post-separation use; prepare the propulsion system and reaction control system engines for their first use ("vent and prime"); detumble from LV tip-off rates; deploy the solar arrays to begin power generation; execute first articulation of the solar arrays, and finally conduct sunsearch and sunpoint activities (find the sun and point the High Gain Antenna (HGA) towards it, shading the spacecraft from sun exposure and orienting the antennas in a known configuration). Once the Spacecraft has completed these activities, it is in a power-positive, thermally safe, and communications-enabled state, where it can safely wait for ground interaction. However, the operations team on the ground must quickly establish two-way communication (if not already established) and begin orbit determination to prevent difficulty in finding the spacecraft on subsequent ground station passes.

Checkout and Cruise—After ground contact is established, the mission performs a series of checkout activities. Several of these must occur prior to the first trajectory correction maneuver (TCM), which could occur as early as launch plus 7 days. One of these is the ICEMAG boom deployment, which is conducted early in the mission due to thermal constraints. While a trajectory with a Venus flyby would mean longer warm conditions, flying direct to Jupiter would expose the Flight System to an increasingly cold environment shortly after launch. With multiple LV options, the mission must be designed for the worst cases of both.

Additional checkouts and calibrations, such as instrument health checks and mass properties calibrations, will take place after the first TCM and within 10 weeks after launch. No later than 3 weeks after launch, the attitude control of the spacecraft will be switched from thrusters to reaction wheels and remain on wheels throughout cruise (except during wheel biasing and maneuvers). This significantly reduces the fuel required by the spacecraft's reaction control system. In order to reduce battery capacity fade during cruise, the mission operations team will allow the battery to drift down to ~60% state of charge. Less than 5% of the battery is necessary to recover from off-Sun faults during cruise, so ~60% state of charge is considered a safe limit. In addition to ensuring spacecraft health and safety, periodic maintenance and calibration activities will occur for all systems that require it throughout cruise and major science calibration campaigns will occur every year (regardless of selected trajectory).

The inner cruise phase covers the period where the flight system is less than 2 AU from the sun. The length of time in inner cruise (and minimum distance from the sun) is highly dependent on the LV selection and whether the spacecraft will fly direct or indirect to Jupiter. The indirect route includes a Venus flyby and the mission's driving hot thermal environment. While in this phase, regardless of the selected LV, the flight system must continue pointing the HGA towards the Sun, where it acts as a sun shade for thermal protection. However, this orientation significantly reduces telecom capability, as the low-gain and fan-beam antennas must be used for communication with Earth instead of the HGA. Once beyond Inner Cruise, there are no thermal restrictions on spacecraft attitude.

Jupiter Orbit Insertion (JOI)—Both trajectories bring the flight system to Jupiter, where JOI occurs as the mission's second critical event. The project plans to conduct this activity autonomously, including the ability to restart the maneuver in the event of an anomaly. Once the operations team has established that the in orbit, the team may choose to execute a stored sequence of unique science observations and instrument decontamination activities.

JOI is an approximately 900-m/s delta-V event that must happen near perijove to capture into the initial Jupiter orbit, and must be accomplished within a constrained timeline. The eight bipropellant engines will provide the thrust for the JOI burn, which will execute for approximately 6 hours. Because of the time criticality of JOI execution, it must be 'fail-operational' and robust to single faults during the event. An autonomous flight software behavior and a unique system mode will provide this functionality. Clipper will be using high level behavior that directs standard lower level behaviors, while also providing the fault recovery functionality. This approach was chosen since the JOI activities are not unique from other delta-V events except for the capability to restart autonomously. This software behavior will be activated long before the event and provide the functionality to walk the system up to the appropriate configuration to execute the delta-V. Fault management will have the ability to pause the JOI behavior to correct any fault that prevents proper execution of JOI, but then hands back control to the JOI behavior to continue executing the remaining delta-V. The behavior architecture and scenario were peer reviewed prior to the project PDR. Near term work will focus on fault scenarios and details of interactions with fault management.

Tour—With a successful insertion into Jupiter orbit, Europa Clipper begins the Tour phase, in which the mission performs flybys of Europa and acquires sufficient data to meet mission science objectives. In total, there are over 45 flybys of Europa during the Tour, most of which have closest approach altitudes between 25 and 100 km. Tour is divided into four sub-phases that each serve a distinct purpose:

- Transition to Europa Campaign 1 (TEC1): pumps down the orbital period through a series of four Ganymede flybys so that the spacecraft arrives in a 4:1 resonance orbit with Europa. The duration of this sub-phase is about 1.4 years.

- Europa Campaign 1 (EC1): consists of a ~1-year series of low-altitude flybys of Europa's lit, anti-Jovian hemisphere (Europa is tidally locked to Jupiter and thus the same side always faces away from Jupiter).

- Transition to Europa Campaign 2 (TEC2): uses a series of Callisto flybys to reshape the spacecraft orbit to enable the 2nd Europa Campaign. TEC2 has a duration of about five months.

- Europa Campaign 2 (EC2): provides a of sequence of low-altitude flybys of Europa's lit sub-Jovian hemisphere, and lasts about 10 months.

The Ganymede flybys during TEC1 provide an opportunity to demonstrate an encounter in the Jovian environment and make final adjustments or address any anomalies prior to the first Europa encounter. The nearly 500-day duration of TEC1 also provides ample time and resources (e.g. power and data) to perform calibrations, ready the instruments for science acquisition, and conduct early investigation of Europa, albeit at a higher altitude than in the following campaigns. By the end of EC1, the high volume of science data acquired during each encounter, combined with limitations on data return capability, may result in a significant amount of on-board data waiting to be transmitted; however, TEC2 provides a reprieve from the high levels of data accumulation found during Europa flybys and thus gives time to return any data carried over from previous flybys.

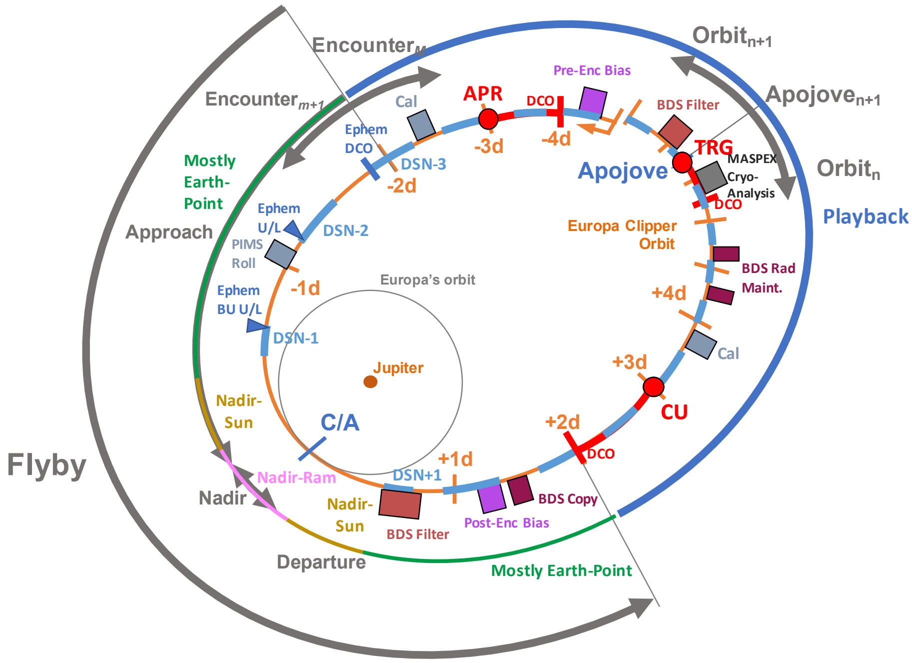

Figure 6: Phases of a Europa Encounter

Flybys and Playback—As described in previous papers, a Europa encounter consists of a "Flyby" (time closest to Europa when the vast majority of science data is collected) and then a "Playback" phase, which ends at the start of the next flyby.

During Tour, the time between 2 days before and after closest approach is known as the flyby period and can be further broken down into 3 parts: Approach, Nadir, and Departure (see Figure 6**)**. During Approach, instruments begin to prepare for and make episodic observations of Europa, and the operations team will have a final uplink opportunity to make small parameter and timing adjustments for the upcoming flyby sequence. The Nadir phase consists of the 4-hour period surrounding closest approach where the spacecraft will maintain a nadir attitude and all instruments will simultaneously acquire data. Departure activities will generally mirror those found in Approach with differences driven by lighting conditions and the need to prepare for data playback. The Playback phase, which follows Departure and ends 2 days prior to closest approach of the next targeted flyby, provides time to downlink the data acquired during the previous flyby back to Earth via the Deep Space Network (DSN). 34-meter Ka-band downlink passes are scheduled at roughly a 50% duty cycle during Playback.

Rapid cadence of targeted science flybys—A 14-day average spacing between flybys of Europa drives the need for numerous, high-frequency overlapping activities on the flight system and on the ground to support science data collection and spacecraft health and performance: 3 orbit trim maneuvers, sequence development and verification processes for all 9 instruments, updates to final flyby ephemeris, engineering activities, instrument calibrations, and control and coordination of downlink prioritization. Use of automation and simplification of ground processes to limit sequence development-to-execution ratio to 1:1 or better can help reduce the overlap of operations tasks during each encounter.

Update on Data Management and Margin—Early in 2018, analyses of the Tour revealed that the current operations plan did not yield the recommended project data margins. A Tiger Team was established to address the issue. The team found that additional margin, sufficient to meet project recommendations, could be obtained by refining instrument data volume estimates and updating the encounter scenario to switch downlink rates more frequently during Ka-band passes during the playback phase. Moreover, additional strategies such as scheduling additional DSN contacts or arraying DSN stations (a strategy which results in higher bandwidth than a single station's capability) can be employed if necessary to recuperate margin.

Disposal—When a mission has the potential to impact and contaminate a body like Europa, the mission must be definitively concluded in a safe manner before the depletion of resources threatens the ability of mission operators to control manner of disposal. The most desirable targets for disposal, aside from being acceptable locations to crash a spacecraft from the Planetary Protection perspective, are those that are most straightforward to reach. For Clipper, Mission Navigators have determined that Jupiter, Io, Ganymede, and Callisto are reachable with the least probability of impacting Europa. Ganymede and Callisto are always easier to get to than Io or Jupiter (from the standpoint of time, delta-V, and TID), and so all of Clipper's recent tour designs have targeted those two bodies. However, given the

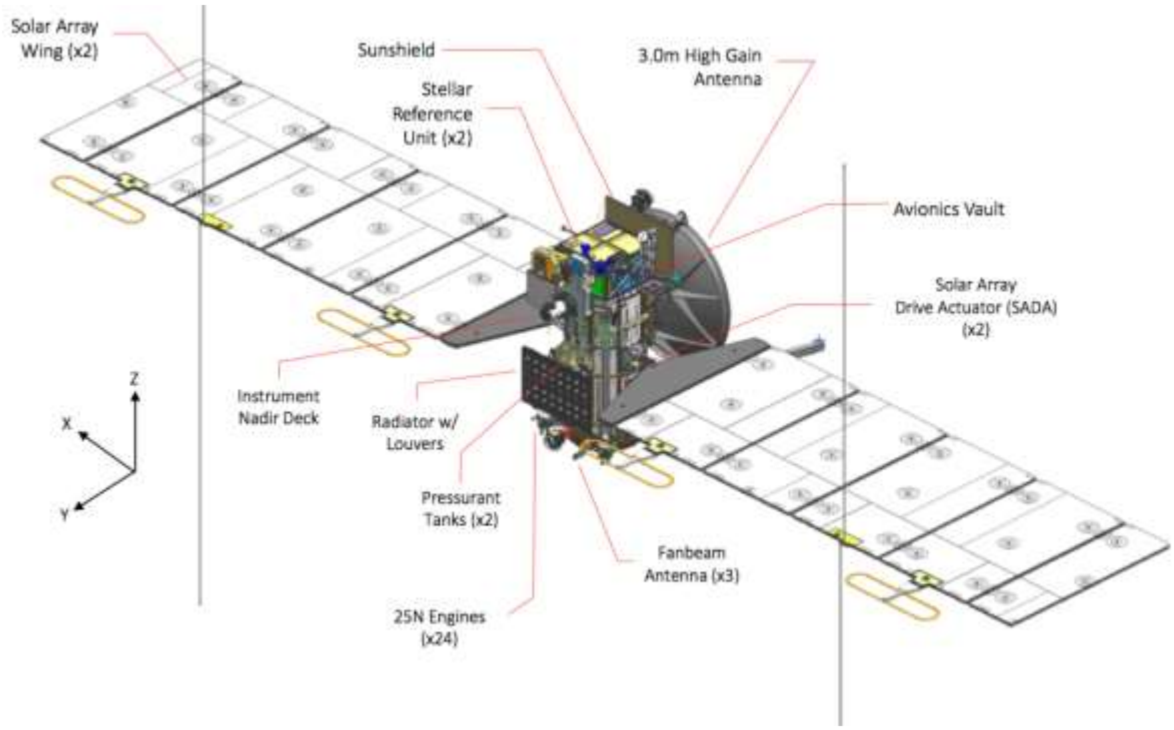

Figure 7: Europa Clipper Mechanical Configuration

potential that Ganymede and Callisto could also possess subsurface oceans, NASA's Planetary Protection Officer has requested that Clipper use Jupiter as its disposal target. Although this request will result in a longer disposal phase with at least one additional Jovian moon flyby to set the spacecraft up for Jupiter impact, the spacecraft design and nominal mission scenarios will likely remain unaffected by this change.

4. FLIGHT SYSTEM OVERVIEW

By the end of the Preliminary Design phase, the flight system architecture, modules, and assemblies are defined and interfaces between components are identified and iterating under configuration control. The traditional flight system architecture provides clear separation between Spacecraft and Instruments with well-defined interfaces and agreements. However, this is not always realistic due to particular instrument accommodation needs and design aspects, and those cases present interesting technical and organizational challenges. This section will describe updates to the Spacecraft and its subsystems in the last year, in giving particular attention to the payload accommodation challenges, compromises, and solutions that exist for this mission, and areas of new focus across the flight system.

Spacecraft Summary

The overall design of the Spacecraft (solar powered, 3-axis-stabilized, pumped-fluid-loop thermal control) has not changed since the description in the previous paper [4].

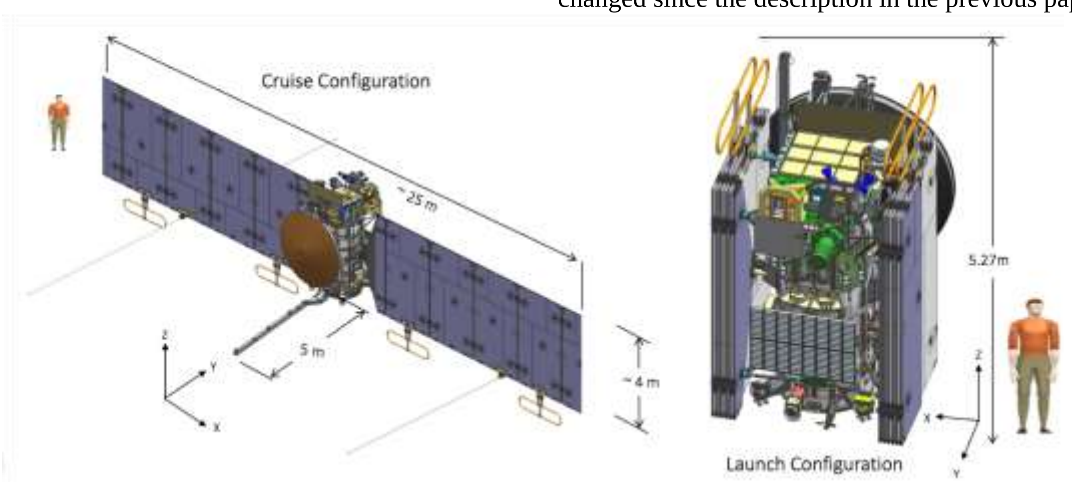

Figure 8: Europa Clipper Cruise Configuration and Launch Configuration

Table 1: Europa Clipper Properties

Statistics
Mass: 2670 kg
336 Ah Battery (End of Mission (EOM))
102 m2 SA area → 700W EOM
5.3 TB Downlink capability
Height: 5.27m
Width: 30.5m (deployed solar arrays)
Thermal: 0.65AU to 5.6AU; (Arrays: +100 to -238C)

As described in [3][4], the spacecraft is physically organized into three modules: Avionics (AVS) Module, Propulsion ("Prop") Module, and Radio Frequency (RF) Module. The functional subsystems cross-cut through the modules. The module organization can be seen in Figure 9. Red elements are thermal hardware that is not technically part of the Avionics Module.

AVS Module: Delivered by JPL, this includes the radiation vault, nadir platform, secondary structure, and thermal loop tubing. (Blue in Figure 9)

RF Module: Delivered by APL as an assembled unit, this module includes the HGA, Medium Gain Antenna (MGA), fanbeam antennas (FBAs), Low-Gain Antennas (LGAs), and other telecommunications hardware. It contains its own radiation vault for its electronics. (Yellow in Figure 9)

Prop Module: Delivered by APL as an assembled item, this module contains the propulsion system hardware, mechanical accommodation, power switching, deployment services, and solar array articulation. It interfaces to the AVS Module, RF Module, ICEMAG boom, launch vehicle adapter, radiator, and includes the solar arrays. (Green in Figure 9)

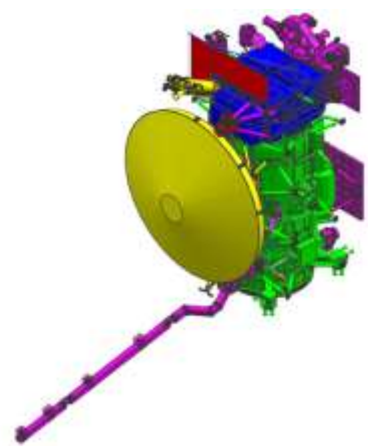

Figure 9: Clipper Spacecraft Color-Coded by Module

Mechanical Design Updates

The most significant changes to the mechanical design in the last year include updates to the Solar Array (SA), vault configuration, and ICEMAG boom.

In-House vs. Vendor-Built Solar Array—Delamination issues observed with the original solar array substrate led the project, after a major trade study, to move to a procured approach for the Solar Array (SA) by partnering with Airbus Defense and Space Netherlands. The decision to buy rather than make presented several technical benefits, including the subcontractor's experience in building Solar Arrays and use of their heritage components (e.g. lower-shock release mechanisms). While the transition was neutral from a mass standpoint, the new array design came with impacts in several other areas: mass distribution and inertial properties; electrical interfaces, configuration of SA restraints, and spacing of REASON antennas. The previous design (4.5-panels with cells on the yoke, included in the A7 configuration discussed in [4]) was thus replaced by a new configuration consisting of 5 panels with unpopulated yoke (see Figure 10). This change, combined with an increase in the spacecraft power demand at end of mission (700W), required increasing size of the panels. Furthermore, the first modes (in-plane, out-of-plane and torsional) of these new arrays are being analyzed to address the pointing stability needs of the Guidance, Navigation, and Control System. The project is now moving forward with this new vendor-built configuration that was the subject of the Solar Array PDR in September 2018.

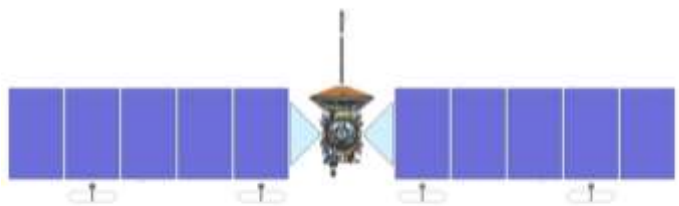

Figure 10: 5-panel Solar Array configuration at Project PDR

Solar Array Articulation—Another important milestone was the selection of the Solar Array Drive Actuator (SADA) vendor. This device contains a stepper motor and a gear box to articulate the Solar Array during the mission. The arrays are articulated for a number of reasons: towards the sun to increase power generation, off-sun to decrease solar heating, or for science reasons. The SADA monitors the Solar Array angle measurement via potentiometers. The SADA is also the point where electrical connections (power from the arrays, telemetry, and REASON power and data) pass from the solar array structure to the Spacecraft. This presents challenges: because the SADA has finite pass-through volume, the number of electrical services going between the spacecraft and solar arrays must be carefully monitored. Additionally, any harnessing going through the SADA is affected by the articulation of the solar arrays, and must not be damaged or impacted by that motion. The SADA and electrical connections that pass through it are depicted in Figure 11.

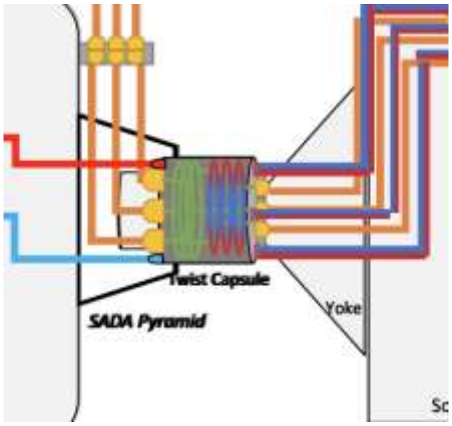

Figure 11: Cartoon of Wires Passing Through the SADA Twist Capsule

Vault Configuration Update—Updates (in Rev. B and Rev. C) have been made to the A7 vault layout initially baselined for the Flight System PDR, in order to address new requirements and to improve access during System Integration and Test (SI&T). Specifically:

- the vault height was reduced to facilitate access to the bottom of the vault
- the thermal pump housing and the pump electronics were moved to separate locations to allow access
- the selection of a different Inertial Reference Unit with optical cubes required specific visual access.

Other design changes were made including a reduction of the –Z Panel thickness (the "floor" panel) as well as adjustments of the vault cable penetrations for instrument electronics boxes before placing them under change control. This revision of the vault configuration (as seen in Figure 12 and Figure 13) led to an overall increase in Current Best Estimate (CBE) vault mass by around 6 kg.

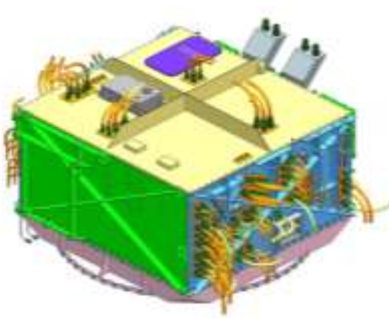

Figure 12: Vault Configuration (Closed)

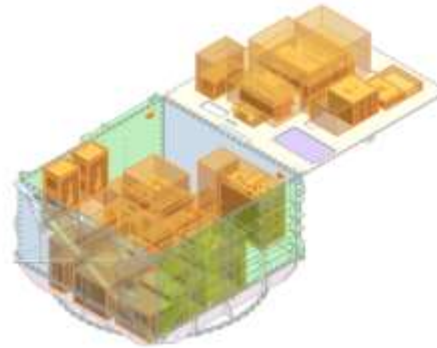

Figure 13: Vault Configuration (Open)

Magnetometer Boom—The ICEMAG instrument includes an array of four magnetometer sensors located along a five-meter boom mounted to the base of the Propulsion Module below the HGA. This boom is provided by the Mechanical subsystem and is designed to meet strict requirements regarding magnetic cleanliness, position and sensor spacing. Before deployment, the boom is stowed behind the HGA and the Solar Array. The deployment sequence brings it around the HGA and down in an arc to its fully deployed position, shown in Figure 14. The magnetometer boom configuration has evolved from a 3-hinge design, to a dual-hinge design with a Stellar Reference Unit (SRU) mounted on it, to the current single-hinge baseline design without SRUs. A single-hinge configuration not only reduced the boom mass by ~5 kg (CBE) but also offered a more robust, simpler implementation and deployment sequence while also reducing the pointing uncertainty of the ICEMAG instrument. The single-hinge design improved, but did not fully eliminate, obstructions of the Solar Array and the REASON antennas during boom deployment. Additional challenges associated with the boom include meeting the ICEMAG sensors' allowable flight temperatures, minimizing leak from the spacecraft to the boom, and with routing the harness across the hinge/along the boom. These issues have been addressed by a Tiger Team formed in the fall of 2017 and that will continue during phase C.

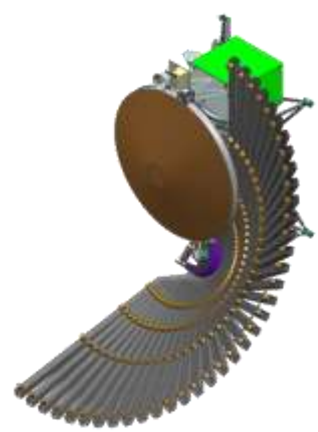

Figure 14: Magnetometer Boom Deployment Sweep (Single-Hinge Design)

Ongoing work—Changes under consideration include the addition of isolators to the Reaction Wheel Assemblies (RWAs) to protect them from the shock and random vibe environments associated with launch. These isolators (as illustrated in Figure 15) are also expected to help mitigate the effects of jitter and microphonics.

Figure 15: Reaction Wheel Assemblies with Isolators, on a Proposed Pyramid-Shaped Support Structure

Spacecraft-Payload Accommodation Updates

Accommodation of the payload on the spacecraft vehicle requires iteration and compromise in order to achieve the mission science goals while minimizing mass and design complexity on both spacecraft and instrument sides. A number of payload accommodation challenges have been resolved since the publication of the previous paper through collaboration between the Spacecraft subsystems, Flight System, Payload Engineering, and Instrument teams.

Instrument component locations—Iteration on the precise location of instrument components has continued with the goal of resolving remaining field of view impingements and optimizing the harness design. The chief concerns are minimization of cable lengths within science parameters (for mass savings and science performance) and reduction of the number of harness segments (segmentation negatively affects science performance). Harness optimization is also addressed by the design of the harness routing through the spacecraft structure. While the instruments build their cables, the routing of the cables is done by the Spacecraft within the length, segmentation, bend radius, and environment requirements of the instruments.

Instrument purge—In order to avoid contamination of instrument optics and sensors before launch, sensitive regions of some instruments must be flooded with nitrogen gas during SI&T and up to launch, with only a small amount of time "off-purge" allowed. The requirements for nitrogen purity, flow rate, and maximum time without purge gas flowing have been negotiated for each instrument. The purge system itself spans ground and flight; instrument designs include purge hardware which connects to purge tubing that is routed on the spacecraft similar to electrical harnessing or fluid loop tubing. This purge hardware is connected to a nitrogen supply system on the ground. The parts of the system attached to the spacecraft and instruments fly with the mission, even though they are not used after launch. The entire purge system is a joint deliverable of the Thermal Subsystem, Mechanical Subsystem, Instruments, and SI&T.

Instrument Electrical Interfaces—The Electrical Systems Engineering Team has been working with the instrument teams to confirm all Spacecraft-Instrument electrical interfaces and capture their detailed specifications in the Instrument-Spacecraft ICDs. Review of the size (current limit) and intended use of the power switch services allocated to the payloads is underway. Another area of electrical interface review is the quantity and correct location of Spacecraft-read Instrument temperature sensors (critical both for management of Spacecraft electrical services, for the correct thermal control of spacecraft-operated survival and operational heaters located on instruments, and for continuing progress in the detailed design of instruments by the Instrument teams). Additionally, the bundling of instrument electrical signals is being updated in order to reduce the number of vault penetrations while avoiding unacceptable signal interference.

Thermal Interfaces—Nadir Deck temperatures and gradients are sensitive to heat transfer to and from instrument sensor heads. To ensure appropriate interface accommodation for the instruments and SRUs, the Thermal team has worked with the instrument teams to determine limits on heat transfer, capture those limits in interface requirements, and ensure that the spacecraft-instrument interfaces have the thermal conductance and isolation properties required for the thermal designs to close on both sides of the interface.

Interface formalization—Interface definition has matured through further work capturing the definitions of Instrument behaviors and expected Spacecraft responses via development of Instrument control requirements and the capture of Instrument functionality and behavior in Instrument Functional Description Documents (FDDs). All Instrument-Spacecraft ICDs have been baselined and a second round of revisions has begun to incorporate the updates and refinements to the instrument designs and more detailed knowledge of the interfaces.

In addition to broad payload accommodation topics that affect all instruments, updates have been made to the unique Spacecraft-Payload interfaces of several instruments:

EIS—The EIS instrument design originally included cross-strapping between its two Data Processing Units (DPUs) to allow either DPU to control either camera. While this provided partial redundancy, it did not provide full redundancy, as the camera assemblies outside the vault are single string. The benefits of removing cross-strapping (reduction in the harness mass by half and a simplified design) outweighed the risks, so the cross-strapping between the two DPUs was removed.

Additionally, the EIS gimbal design was updated to be lighter and shorter in profile, and to use a single frangibolt rather than two frangibolts. This meant one launch lock instead of

Figure 16: MISE Mechanical Updates

two, and simplified the integration of EIS with the Spacecraft overall.

REASON—A key area of payload accommodation efforts has been the interface between the REASON Instrument and the solar array (on which REASON is mounted). The decision was made to change to a vendor-built solar array (as discussed in the Mechanical Design Updates section of this paper), resulting in updates to the RF grounding schemes. In addition, REASON was also able to remove the survival heaters on their matching networks after the matching networks passed survival qualification. The benefits to this design update include power savings and improved electrical resource and power margins. The removal of heaters also resulted in improved torque margin across the solar array hinge lines due to the reduced number of wires across the hinge lines. The number of wires through the SADA was reduced as well. Additionally, REASON baselined the use of silicon dioxide coaxial cables, which simplified the design; this type of cable is qualified to the required temperatures and radiation levels, has a more predictable and well-understood performance over mission temperatures than other options, and does not have internal electrostatic discharge concerns. Finally, REASON also added launch restraints to the VHF design. Made of a non-dielectric material, these latches hold the VHF antennas during launch and then release post-launch, at which point the vacuum provides electrical isolation.

MISE—The design of the mechanical interface between the MISE instrument assembly and the Spacecraft was modified to address a number of requirements and constraints that were not adequately met by the previous design. Originally a three-point bipod, the structure supporting MISE is now a larger single frame that meets the MISE requirement on allowable motion between the cryocooler structure and the optics bench assembly, exceeds the frame stiffness goal, and meets the constraints regarding accessibility of the MISE bolted interfaces and Mechanical Ground Support Equipment (MGSE) during SI&T. The single frame design also reduces risks to the MISE hardware during the Instrument and System I&T campaigns. While the previous design required both the MISE FPIE to be installed separately from the optics bench assembly and the MGSE to directly support the optics bench, the new single frame design supports all of the MISE hardware in one structure which reduces the number of pieces to integrate and removes MGSE from the critical path of the detector integration.

MASPEX—Accommodation of the MASPEX instrument included a number of changes since the previous paper. The positions of MASPEX and SUDA (both sensor heads and electronics boxes) were swapped in order to resolve the obstruction of the SRU keep-out-zone by the MASPEX thermal blankets, and to provide robustness against a number of uncertainties including mounting structure and volumetric growth for microphonics isolation. The MASPEX mirror pulser control circuits were also moved from the external instrument assembly to the MASPEX electronics box inside the vault, and additional shielding was added to the pusher/extractor circuits in their original location outside the vault. This solution addressed radiation hardness and reliability concerns related to the original pulser circuit design. The MASPEX cryocooler and radiator were moved to the top/front of the MASPEX assembly, closer to the to the inlet, in order to address the cryotrap not meeting the conductance requirement. The MASPEX mounting structure design was transferred to the MASPEX team at Southwest Research Institute and integrated with the instrument, reducing the number of handoffs between different institutions and thus reducing the probability of error. The mounting structure itself was also redesigned to address concerns regarding cantilever loads on the honeycomb plate.

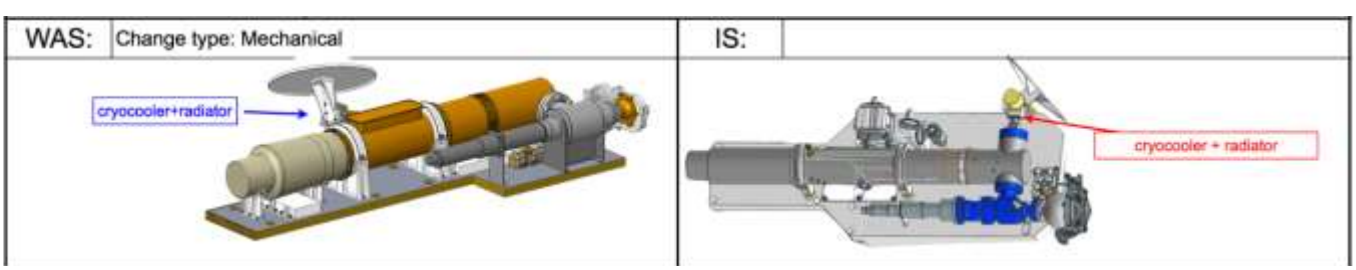

Figure 17: MASPEX Mechanical Updates

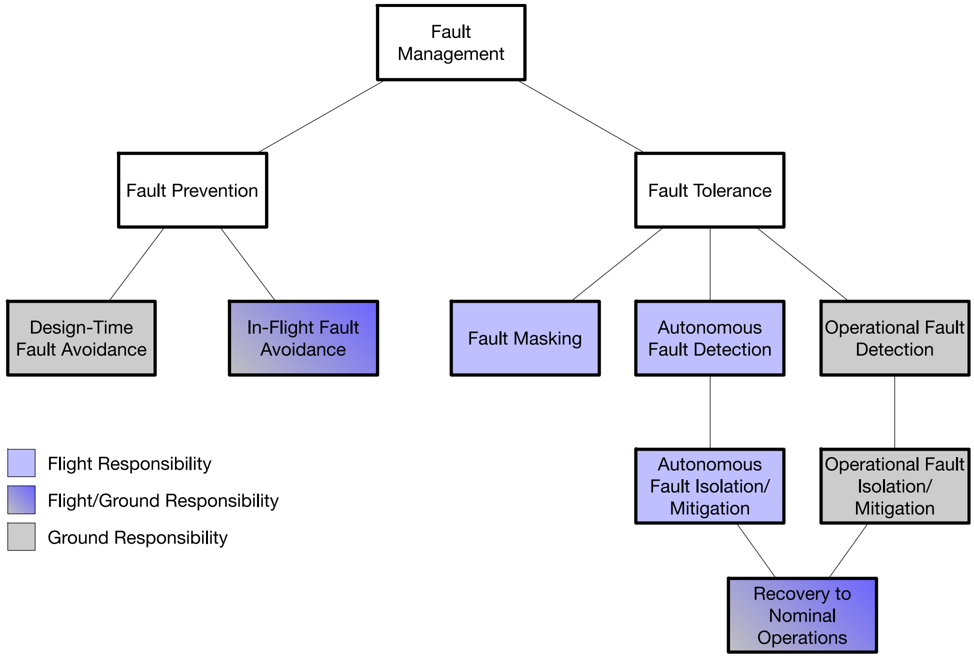

Figure 18: Fault Management Architecture

Finally, an additional power service and eight temperature sensors were added. The single Spacecraft power service that controlled an array of heaters (which performed operational and non-operational heating) on MASPEX was split from one controlling switch into two, which allows for two thermal zones instead of one and thus more efficient energy usage. The addition of eight temperature sensors makes sure that the control scheme for the thermal zones includes one pair of temperature sensors per zone, which simplifies testing and decouples the zones in the event of a failure.

ICEMAG—To address tight margins on its printed wiring board area, ICEMAG increased the height of its electronics box in the vault. Additionally, ICEMAG increased its required Spacecraft power services by four. Originally, all five ICEMAG non-op heaters (four on the sensors, one on the fiber cable, intended to keep ICEMAG sensors and cable above survival temperatures when not operating) were powered by a single switch. Designating a separate switch for each heater avoided impacts associated with design and test of a new and complex heater management algorithm and increased efficiency by allowing for local (and optimized) control of each individual thermal zone.

E-THEMIS—As the E-THEMIS instrument design was refined, it was discovered that the sensor assembly transmitted launch loads that were too high for the Focal Plane Assembly (FPA), introducing a risk that the FPA would not survive launch. Subsequent iteration on the FPA and optical bench designs resulted in changes to the optical bench dimensions and the addition of four flexures to achieve the desired stiffness. Additionally, it was determined that the sensor assembly operational power dissipation resulted in overheating. To address this issue, a radiator was added to the side of the FPA enclosure; this, along with updates to the sensor assembly's operational temperatures, brought the FPA back into an acceptable temperature range. Altogether, these modifications slightly increased the sensor assembly footprint and volume envelope but achieved a viable design for E-THEMIS that could be accommodated by the Spacecraft.

Future Accommodation Efforts—Future payload accommodation efforts will include the following: identifying opportunities for reduction of vault wall penetrations via combination of signals where feasible; trading the cable lengths required by the current Spacecraft configurations against potential signal degradation associated with long cables and/or multiple cable segments; addition of redundant temperature sensors to instrument thermal control zones for increased robustness; impact assessment of Spacecraft loads on the Instruments, specifically microphonics on PIMS; and further definition of the Instrument accessibility and ground support equipment required to perform Spacecraft integration and test, as well as refinement of test approaches for verification and validation of payload requirements. These accommodation agreements will be captured in the payload requirements, Mechanical Interface Control Documents (MICDs) and Instrument-Spacecraft ICDs. All interface control documents will mature from Baseline to Final as the teams prepare for the Instrument CDRs throughout the first half of 2019.

Fault Management

As described in the previous paper [4], the fault management architecture and design are driven by the Project single point failure policy to provide fault tolerance for mission-critical functions. This involves both hardware design considerations (e.g. redundancy and fault containment regions) as well as

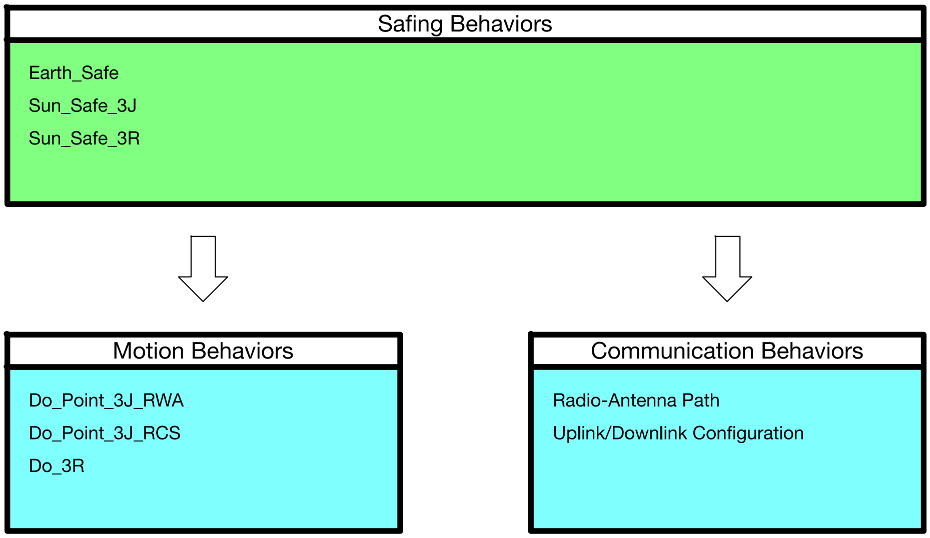

Figure 19: Safing Behavior Hierarchy

flight/ground design responsibility for fault detection and isolation/mitigation.

The fault management PDR was successfully held in May 2018. The overall fault management architecture and design was presented (e.g. principles/requirements and drivers for autonomous recovery) as well as implementation details for system and subsystem fault tolerance and the preliminary software design. This section will describe a few of the PDR topics as well as the path forward to CDR.

Safing Response and Behavior Architecture—The safing response can be autonomously invoked to establish a power-positive, thermally safe, and communicative state in response to a fault. This allows the flight system to be in a safe and predictable configuration for further diagnosis by the ground before returning to nominal operations.

The safing response has to consider both the severity of the fault and mission constraints in order to establish a given safe state throughout the mission. One of the fault management design drivers is the principle to preserve capability in order to minimize disruptions to nominal operations. For example, staying on RWA control (when possible) will preserve consumables and downlink performance as compared to always defaulting to Reaction Control System (RCS) control. However, RWAs cannot be used during the Inner Cruise phase because of thermal constraints. In order to manage these multiple design constraints, the safing response is currently implemented as a hierarchy of behaviors.

The safing response calls a given safing behavior depending on what attitude knowledge and control is available after isolating/mitigating the fault. The safing behavior then coordinates the actions of lower-level motion and communication behaviors to establish attitude targets that are consistent with comm. configurations. For example – earth-pointed attitudes can use the HGA when on RWA control whereas sun-pointed attitudes can only use the MGA or FBAs.

Safety Net Fault Responses—Safety net fault responses are designed to mitigate faults where the symptoms cannot be isolated to a single cause (e.g. due to ambiguous error detections or operator errors). These safety net fault responses have a never-give-up design philosophy that autonomously reconfigure the system (in a tiered approach) in order to isolate/mitigate the fault.

Low Energy/Undervoltage Response: A two-tiered response based on bus voltage that sheds loads and establishes a power-positive configuration. The first tier is a software-based response and the second tier is a hardware-based response that power cycles and swaps redundant hardware.

Command Loss Response: A multiple tiered response based on the expiration of a timer that establishes different comm. configurations. The first tier establishes uplink/downlink on the primary antenna path and subsequent tiers establish different antenna paths on different radios and computers.

Verification and Validation (V&V) Approach—The V&V approach for the fault management design includes the verification of requirements allocated to subsystems as well as the verification of the monitors/responses and behaviors in the software design. The primary venue for fault management V&V is known as "STB1," a hardware-rich dual-string system testbed that allows for the characterization of flight-like hardware/software timing and interaction. In order to perform end-to-end functional tests (error injection to monitor detection and response isolation/mitigation) – requirements were levied on the test venues and simulation support equipment to inject symptoms of faults in a flight-like manner as possible (e.g. injecting user-specified telemetry to engineering models such as temperatures/currents/voltages).

Path to CDR—The path to CDR involves a transition from fault management requirements and analysis products (e.g. fault identification in FMECAs) to the actual monitor/response design in flight software. This involves the construction of a mitigation matrix (a consolidated set of detections/mitigations) and a specification of flight software architecture and patterns. In addition – the effectiveness of the fault management design itself will be evaluated via coverage, timing, and interference analyses. Fail-operational behaviors for time-critical events such as Launch and JOI have unique fault management considerations where the interaction between nominal and off-nominal actions has to be clearly defined and understood. Lastly – a detailed V&V plan will be developed that maps all of the fault management verification activities to software releases while considering schedule and workforce constraints.

Thermal Control

Clipper's thermal management system is designed to maintain safe survival and operating temperatures across the flight system in both the extremely hot conditions near Venus and in the extreme cold of an eclipse near Jupiter. The design consists of a pumped fluid loop that captures waste heat from electronics in the vault and uses it to keep the propulsion system components within their allowable temperatures. As needed, a large group of heaters known as the Replacement Heater Block injects additional heat into the loop when waste heat is not enough to maintain system temperatures.

This architecture has not changed significantly in the last year; work has focused on refining mass and power estimates and allocations; reducing the impacts of vibrations from the fluid loop pumps; minimizing heat loss, and beginning the plans for V&V of the system.

While most spacecraft components are on the thermal loop (which maintains their temperatures), the majority of instrument components (outside the vault) are not. For components off of the fluid loop, thermal control is achieved with traditional software-controlled heating: temperature sensors are read by flight software, which turns nearby heaters on and off according to high and low set points. Over the last year, the team has investigated methods to improve the robustness of these non-loop thermal zones, and proposals to increase the number of temperature sensors (to protect against failures) and removal of the mission's last mechanical thermostats are in work.

Maneuver and Pointing

The primary driving pointing requirements have been constant over the past year. Areas of development over the past year were changing the sun sensor hardware, updating the pointing stability assessment, adding reaction wheel isolators, and improving analysis of contributions to pointing knowledge. The following section describes the work that led to these changes.

During the last year a decision was made to change from coarse sun sensors (CSS) to a digital sun sensor (DSS). Previous analysis indicated that the system would close with the CSS hardware if there was a slight relaxation of accuracy requirements coupled with a refined set of applicable scenarios of use. However, further analysis showed that the CSS had significant limitations on handling reflected sunlight, and overly restricted the design space for usable scenarios. The decision to move to a DSS allowed for a more robust implementation, especially as a key attitude sensor used in safe mode.

Meeting the pointing stability requirements continues to be a challenge. Modeling the system for high frequency disturbances has been challenging because as the system matures, changes in the configuration and refinement of instrument models have had large impacts on the modeled performance. For example, significant design changes to the EIS NAC mounting interface, the instrument nadir platform, and nadir platform attachment have all impacted the assessment process. Additionally, the reaction wheel disturbance model indicated a major non-compliance. A small Tiger Team was formed to address the reaction wheels performance. First, the reaction wheels were not specified to operate in the launch shock environment, and in addition they were imparting too much disturbance at the NAC during operations. To address these issues, an isolation system was incorporated into each wheel. The isolation system removed the higher frequency harmonic concerns from the RWA addressing both the launch environment and the NAC disturbances. Finally, the assessment methodology for the RWA was updated to be probabilistic with respect to wheel speeds, instead of the overly conservative approach which assumed the worst case at all wheels at the same time. We continue to expect the pointing stability requirements to be challenging but believe that we are on a good path with regard to RWA disturbances.

The last item that has been worked heavily in the past year has been pointing knowledge. Three main components that have been worked are: thermal-mechanical distortions, Inertial Measurement Unit (IMU) performance, and operations adjustments. Thermal-mechanical distortion between the SRU and IMU directly contributes to a reduction in pointing knowledge while propagating attitude with the IMU. Detailed investigation into the applicable scenarios and the expected thermal environments has provided a better set of thermal maps. The thermal team and mechanical team have developed a process to quickly apply thermal gradient maps to the FEM and compute distortions between key interfaces. While the SRU-IMU requirement is still not quite met, we are much closer and have higher confidence in the numbers. IMU performance was another area of improvement. When the transition to a hemispherical resonator gyroscopes-based IMU was made, the intent was to operate them with internal heaters off, providing power and energy savings during the science tour. However, further analysis showed that the performance needs would not be met without the internal heaters providing a stable temperature, and the baseline has been updated to include use of power for the internal heaters. The last aspect of reconstructed knowledge that has been updated in the past year is a deeper investigation into the scenarios where we need to propagate attitude using only the IMU. It has been identified that small operational changes could be made that would reduce the duration of times when the SRU was obscured, which reduce the reliance on the IMU performance. There is still work to go on the thermal mechanical distortion analysis, but the pointing knowledge performance is in a robust state.

Propulsion

The propulsion subsystem remains a bi-propellant Earth-storable design that includes two fully redundant sets of twelve 25 N-class engines to perform both delta-V and attitude control maneuvers. Eight axial engines will be used for the delta-V maneuvers, while four engines on the Y-axis facing sides of the Spacecraft will provide roll control. The propulsion subsystem passed its CDR in June 2018 and is ready to start building and assembling the subsystem. The identical titanium propellant tanks, capable of storing 2750 kg of propellant collectively, have also passed CDR and are beginning the manufacturing phase. The tanks take advantage of a Propellant Management Device to manage disturbances caused by propellant sloshing during the mission lifetime. Manual valves have been added to the design of the Propellant Isolation Assemblies and the Pressurant Control Assemblies for purge and leak testing. Modeling work for Active Pressure Control (APC), responsible for managing the pressure in the propellant tanks, has been completed and verified with flight-similar testing. The verification efforts prove the robustness of APC as it minimizes propellant residuals in the tanks to optimize propellant use, prevents vapor mixing, and allows for better performance in maneuvers late in the mission.

As with any space mission, there are a few challenges facing the propulsion subsystem that are actively being worked. The Europa architecture does not use large main engines relying instead on on- and off-pulsing of smaller engines to perform maneuvers. The frequency of off-pulsing is associated with the spacecraft center of mass (CoM) offset in the coordination plane perpendicular to the delta-V engines. A larger CoM means that engines located on the side of the spacecraft of the CoM offset will have lower duty cycles than they would with a more ideal CoM. The off-pulsing of the engines reduces their effective specific impulse (Isp), since the Isp of this engine type is a strong function of engine duty cycle. Collaboration efforts are ongoing between the propulsion and delta-V/attitude control designers to ensure the engine performance is optimized.

Use of an oxidizer as a propellant results in leaching of iron from the stainless-steel components of the propulsion subsystem. The resulting iron nitrate condenses into the propellant as particulates that can, over time, block filters and ultimately result in a failure of the system. Firing the engines frequently flushes out the particulate and can address the issue; however, this strategy requires extra propellant, and engine firings during long periods of engine disuse must be planned. Additionally, these firings do not address the possibility of iron nitrate formed at other critical points in the subsystem where stainless steel exists, such as at pressure transducers (where the concern is a blockage of the venturi flow meter sensing port) and the Pressure Control Valves (PCVs), which could become contaminated due to condensed oxidizer vapor. A new solution has recently emerged: implementation of pre-flight passivation of the propulsion components by exposing them to wet oxidizer at elevated temperature. This exposure forms chromium oxide on the surface of the steel, which stops further iron leaching. Passivation will eliminate the need for engine flushings during the mission and remove concerns associated with iron nitrate formation at the pressure transducers and PCVs [10].

With solutions to these challenges in work, the propulsion subsystem team is focusing on V&V planning and integration into the Prop Module and ultimately the spacecraft.

Power & Electrical

As the spacecraft design has evolved, the power subsystem has maintained its distributed architecture. The main control and distribution hardware element (the Power Control and Distribution Assembly (PCDA)) remain inside the avionics vault while a second set of redundant distribution hardware (Propulsion Module Electronics (PMEs)) are located in their own vaults on the Prop Module. All main primary power for subsystems and instruments is provided via the PCDA. Power for all propulsion and deployment loads routes through the PMEs.

Power and energy needs across the system as well as power generation capability (solar array sizing and wiring impacts) are under review. In an effort to meet system demands, the power servicing capability has expanded to accommodate 300 unique loads, though total load count is expected to be much less. Each power service has been implemented using a high and low side solid-state power switch pair, commanded as one functional entity and collectively known as a switch channel, but most commonly referred to as a switch. The base switch channel ratings are 2A, 2.5A and 5A. Other ratings, namely 4A and 10A, are available and made possible via parallel configuration of two base switch channels, although use of these other ratings lowers the available base switch count accordingly. Additionally, certain selective switch channels may be configurable as safety (breakwire controlled) or critical (auto on) type. All existing system power loads have been assigned switch channels.

System resources (switches, along with other electrical I/O such as temperature sensor channels, etc.) and their assignments are under configuration control, with corresponding margins tracked and reported to project management. New power requests are continually reviewed, with allocation decisions based on a balance of needs (performance, efficiency) and impacts (physical, thermal, future needs, cost, schedule etc.). Potential uses of available

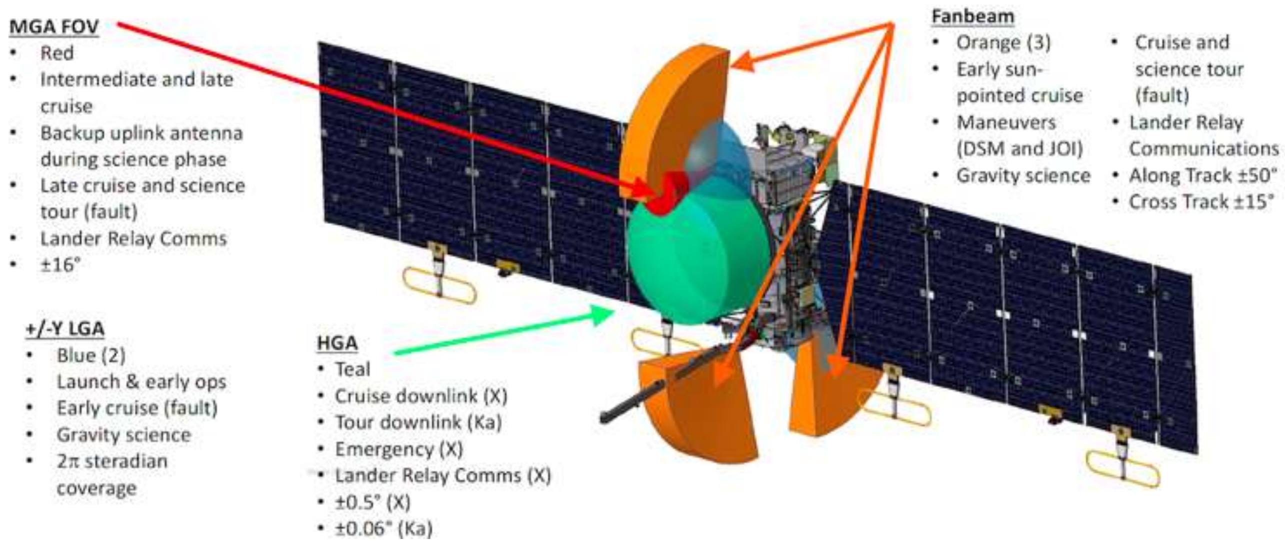

Figure 20: Antennas and Fields of View

switches under review include providing redundant power to critical loads and adding heaters to improve robustness of thermal zone control.

As the system design matures from functional mapping to hardware and channel assignments, the flight system team is working toward the final stages of the physical definition and implementation; conductor quantity, gauge, treatment and shielding.

A Power FDD has been written to provide the flight software development team with functional and performance specification required to manage the power subsystem. The entire electrical system continues to be documented and analyzed, with focus on such areas as grounding to ensure isolation is present where necessary (no sneak paths), voltage drop to verify loads receive the power needed for proper operation at high and low power bus levels, and magnetic cleanliness to aid shielding and filtering decisions. Other electrical reviews planned are expected to provide further input to harness development and thermal isolation impacts.

RF Module/Telecommunications

All telecom functionality on board the Spacecraft is contained within the RF Module, which is designed and built by APL and has high heritage. As described in [4], the RF Module includes a suite of antennas for communicating with Earth and for doing gravity science experiments: a 3-m HGA, an MGA, two LGAs, and three fanbeam antennas.

The RF module also includes an RF panel. The RF panel includes a mini vault to shield the electronics from the radiation environment, four Travelling Wave Tube Amplifiers (TWTAs), RF components including switches, hybrids, diplexers, and the mechanical, thermal, and harness accommodations needed by the antennas, and waveguides that connect to the antennas. The mini vault contains two Frontier Radios and four Electronic Power Conditioners that provide power to the TWTAs.

The HGA, in addition to serving as the primary antenna past the 2 AU sun-distance for both X and Ka-band, protects the Spacecraft from the intense thermal environment of inner cruise by acting as a sun shield from the time of launch until the Spacecraft is 2 AU from the sun. All other antennas are X-band only and are used for communications during all phases of the mission, including launch, cruise, safe mode, and JOI. The RF Module is key for navigation, gravity science during flybys, and possible contingency relay with a potential Europa Lander.

Status—The RF Module PDR in March 2018 was successful and the system is charging forward towards CDR in March 2019. The Frontier Radios have been built; their firmware and software are being completed and tested. Requirements linkages from L2 to L3 and L4 continue to mature, including the verification approach plans. One challenging area is possible contingency relay for Lander. As the Lander project is pre-Phase A, its launch date and trajectory are unknown, which makes planning for relay difficult. A bright spot in this uncertainty is that the Lander Project is expected to also use Frontier Radios from APL, so the Clipper and Lander radio development teams share personnel and lab space. This facilitates planning for and testing relay communications. However, Europa Clipper will still need to test with Avionics to ensure that the relay data can be received and recorded fast enough to meet requirements. A Memo of Agreement between the Clipper and Lander projects regarding telecom support is currently in work.

Use of tones for safemode recovery—to accommodate power and mass limitations on the spacecraft, the telecom subsystem was designed to be as efficient as possible for nominal operations. This is reflected in the use of a low-mass Mars Science Laboratory (MSL) heritage MGA, and a 20W

TWTA. However, this results in hardware that does not provide as much signal strength for off-nominal operations scenarios, particularly spacecraft safemode events, as is generally available for deep-space missions. For certain spacecraft distances and geometries, special tactics will be required to restore communications after a safemode event, whereas typically, low-rate telemetry data (e.g. 10-40 bps) would be continuously transmitted to the ground.

These tactics include “clocking” (moving the antenna boresight around the estimated sun position), as well as a “rotisserie” roll (rotating the spacecraft Y-axis about the sun-line). Clocking and rotisserie modes require ground involvement to monitor and stop the rotation at a position that is optimal for restoring communications, which allows diagnosis of the fault that resulted in the safemode execution. Due to the complex nature of the interaction required, and the long one-way light times, these ground activities could take as long as 22 hours to execute.

Communications during some of these safe mode scenarios will rely on Multiple Frequency Shift Keying (MFSK) tones. These tones provide information at a low rate, about 8 bps. Historically, MFSK tones have been used for critical events that have high dynamics and low signal strength. The Mars Exploration Rover and MSL projects utilized tones for direct to Earth communication during entry, descent, and landing. Juno also used MFSK tones for JOI.

From an operability perspective, this design results in compromised visibility into the state of the flight system, particularly for the critical period immediately following a significant spacecraft anomaly. In addition, tractability is reduced by the need to add more operational steps and decisions before basic information is received.

These impacts to operability were identified, and the Flight Systems Engineering team was engaged to perform a trade study to determine if it was reasonable to make changes to the baseline spacecraft design to improve its operability. Several options were identified that could mitigate these impacts to operability, in particular a higher-power TWTA, as well as a larger, more capable MGA. However, due to the significant negative mass and power impacts these measures would incur, the decision was to retain the baseline design.

While this trade was not decided in operability’s favor, the flight system did make an effort to reduce the impact and risk to operations. One step taken was to re-examine the time required for the spacecraft to restore communications after a worst-case fault, which provides operations more time to perform the steps described above. Another step was to carefully analyze the potential fault scenarios, to ensure that only very low likelihood, non-exempted faults would result in the spacecraft being placed in sun-relative attitude determination modes, thereby reducing the probability that these complex operational activities would need to be executed.

Avionics and Data

The Command and Data Handling (C&DH) subsystem (commonly referred to as “Avionics”) is responsible for active control and health monitoring of the Spacecraft. It receives and executes commands, stores and executes sequences, collects and stores telemetry, and manages the command and data interface with engineering and science clients. It is also responsible for running fault monitors and responses, autonomous behaviors, and the orchestration of system modes.

C&DH Hardware—The subsystem hardware consists of two redundant strings (a prime and a backup), each string consisting of two core components, a Europa Compute Element (ECE) and a Remote Engineering Unit (REU). The ECE will provide a radiation hardened RAD750 main flight computer, while the FPGA-based REU will be used for collecting and digitizing Spacecraft Engineering Telemetry. ECEs and REUs are cross-strapped so that each ECE can communicate with each REU, which improves robustness. Furthermore, each ECE and REU will come with associated input/output cards and non-volatile memory (NVM) to meet the Flight System data interface, processing, and storage needs.

On-board, the Europa Clipper Spacecraft Intercommunications Network will support a set of point-to-point high-speed data interfaces (e.g., SpaceWire, up to 200 Mbps capability) to Spacecraft Engineering Subsystems and Payload Instruments. An additional set of Universal Asynchronous Receiver-Transmitter (UART) (up to 115 kbps capability) for low-rate Payload Instruments and RS-422 interfaces will be employed for low-rate Spacecraft Subsystem needs. A shared 1553 bus, a set of point-to-point Remote Serial Bus (RSB) and Low-Voltage Differential Signaling (LVDS) interfaces will also be used to provide robust interface connections among a distributed set of spacecraft engineering devices. For data collection during Europa flybys, all Payload Instruments are powered on and collect Payload Data at the same time; that is, in addition to the execution of the Spacecraft functions to support the flyby activity. This requires that flight system command, data handling, and the intercommunication network are able to concurrently collect and store data (for later sorting and processing) at a peak combined rate on the order of a GiByte per second. Of note: a set of supplementary flight system interfaces will be installed to support Ground-based testing (e.g., GSE), and on-pad and launch operations.

Compared to historical JPL deep space missions, the Europa Clipper mission will carry significantly more non-volatile data storage capability to meet its unique science and engineering data storage needs. Each ECE will have a redundant Bulk Data Storage (BDS) device designed to hold at least 512 Gibibits of Payload Data (volume required at the end of mission, after accounting for memory degradation over the expected life of the mission). Moreover, each ECE will also hold another 146 Gibibits of non-volatile memory for Flight System engineering use (e.g., FS Engineering data storage, event and activity logs, control files, and configuration and parameter tables). There are several drivers for needing the rather large memory volumes: a) the Flight System design is expected to carry large data storage margins, b) during each Europa encounter, the FS may collect and store on the order of 100 Gibibits of Payload science, calibration, and housekeeping data, c) the Flight System is being designed to collect a rich set of engineering telemetry for downlink for Ground-based monitoring and trending, and lastly, d) following data collection, the scheduled RF communication links to Earth may not support the downlink of all of the stored data before the subsequent Europa flybys. As such, the BDS and Engineering NVM must contain a robust amount of storage capacity to allow the interim accumulation and integrity of collected data, until the data can be processed (e.g., sorted and when applicable, compressed) and downlinked by ground-specified priority.

Software Architecture—Europa Clipper Spacecraft's Flight Software (FSW) architecture will use software space and time partitioning. Here, space partitioning refers to the division of memory into isolated activities that enhance the execution robustness of critical software functions (e.g., fault protection) without interruptions by lower priority activities. Time partitioning allows the operating system to schedule multiple threads and processes in order to enhance the predictability of the software execution over time. The space partitioning concept allows for construction and operation of independent components. Time partitioning improves the determinism of FS behavior by fixing the execution schedule of key software activities. This architecture also provides increased robustness, as failure in one partition does not propagate to other partitions. Additionally, the FSW architecture allows for autonomous inter-string communications, and state and parameter updates to enable, through resets, quick transition from prime to backup string.

Software and System Functionality and Behavior— The ECE FSW will be running on the RAD750, with 1 GiByte of volatile memory. FSW will provide system-level commanding, autonomous sequencing (i.e., control programs execution), behavior execution, and Health and Status monitors and responses. Additionally, the FSW provides: data collection, storage, and management (e.g., data integrity, compression); intercommunication network management; time management and distribution; thermal monitoring and control; and execution of guidance and control algorithms. The FSW will also contain monitor and response functionality necessary to implement a significant part of Fault Management for the flight system. Moreover, it will provide uplink and downlink processing services geared at handling the processing of commands and files sent by the Ground and managing the downlink of real-time and recorded flight system engineering data. This engineering data provides visibility into flight system operations, onboard flight system health and status, and any Spacecraft Ancillary and/or Payload Instrument Telemetry necessary for science data processing.

To minimize complexity of data interfaces and management, the Clipper mission uses standards whenever possible. For example, the mission will employ CCSDS File Delivery Protocol (CFDP) for automated acknowledgement and retransmission, both on uplink and downlink. Data formatting and time management also employ CCSDS standards. Moreover, SpaceWire, UART, and 1553 interfaces are also standardized.

5. Project Status and Updates

Reviews and Milestones

The overall approach to the Clipper review process was described in [4]. Since that writing, the project has completed all of its subsystem and instrument Preliminary Design Reviews (PDRs), the Mission System PDR, and the Project PDR. However, there is one last PDR type review scheduled: the Integrated Wing Review.

As described in [3] and in previous sections of this paper, REASON accommodation on the solar array is a technical challenge with the mechanical, electrical, and thermal coupling being the primary drivers of complexity. Accommodation is also a programmatic challenge, and decomposing the technical aspects according to traditional project organization (independent engineering domains and separation of instrument and spacecraft concerns) was found to be inefficient and often confusing. The Project quickly realized that the integrated solar-array-REASON system needed to be treated as just that; a distinct system. Accordingly, the Europa Clipper project initiated an Integrated Product Development Team led by the project Chief Engineer and mainly staffed with REASON and Solar Array Team members. While both the Solar Array and REASON have completed held individual PDRs, there were a number of topics that deserved focus at a dedicated review of the combined system, including integration and test and verification and validation. The Integrated Wing Review will occur in early 2019.

The Europa Clipper Project PDR was held in August of 2018, and as it did in preparation for Flight System PDR, the project declared a baseline freeze and reviewed the design to ensure it was internally consistent before the Project review. A resynchronization was necessary due to the length of the PDR season, with the bulk of the reviews spread out over nearly a year. Since the baseline had evolved slightly over the course of the PDR season, different systems had slightly different baselines as the backdrop for their own reviews.

The resynchronization activity did not identify any significant discrepancies, and the main follow-on activity was a reconciliation of the power, mass, and data resource allocations. This involved gathering all of the resource current best estimates and uncertainty values from each of the Instruments and Subsystems, and relocating resources from the project reserves, thus allowing each of them to have a clear (and hard) limit that their deliveries must meet. In addition, the project was able to present an internally consistent report on the key technical margins at PDR.

All project systems, subsystems, and instruments are proceeding with their detailed designs in preparation for their CDRs, the majority of which occur from December 2018 to November 2019. Due to the long lead items in the propulsion subsystem schedule, that subsystem already completed its CDR in June 2018 and has already started fabrication on those long lead parts (the most notable being the propulsion tanks).

6. VERIFICATION AND VALIDATION

In the initial design phase of the mission, the focus of the systems engineering effort has been on the development of requirements and definition of a preliminary design to meet those requirements. As the project progressed towards PDR, this was augmented with initial planning for verification and validation. V&V is the process of ensuring that the design will comply with the requirements placed on it (verification) and that the as-built system will meet mission objectives (validation). To create a robust V&V program, three aspects of the Europa Clipper project have been emphasized in this planning stage: breadth, depth, and lifecycle [11].

Clipper's V&V program will achieve breadth through its application to all aspects of the mission's function and performance, including not only the hardware and software that will be launched on the Flight System but also functions such as the ground data system, science and data process algorithms and software, and operational processes, including operator and science decision making. To coordinate this breadth, the project uses two key mechanisms: a Project V&V Plan, to describe V&V policies and processes, roles, and individual products used for V&V across the project, and the assignment of engineers to coordinate the planning and execution of V&V in each major system, e.g., spacecraft, mission system, and between systems.

The project will achieve depth in the V&V program by following the traditional "V" approach, where the right-hand branch of the "V" represents the flow-up of the results of V&V performed on components, and then subsystems, before testing at the system level. This effort is also coordinated by the Project V&V Plan, which includes guidance for processes that will be applied at all levels of integration. For example, Europa Clipper will perform tests in the most flight-like way possible, with the intent that the mission will be operated within the parameters tested on the ground. This is known as the "Test As You Fly" (TAYF) principle. Exceptions to this principle will be identified and tracked at all levels of integration and reported at milestone reviews so that related risks can be mitigated by higher-level tests, or recorded as part of the project risk management process.

The V&V program is applied across the full project lifecycle, from phase A through phase E. Europa Clipper is no different from other projects in that the emphasis in phases A, B and the first part of Phase C is on defining policies and requirements on the process, on writing plans, and on creating features that will facilitate later V&V, e.g., including verifiability information in requirements sets. This planning continues until the System Integration Review (SIR), by when the main focus will have shifted to executing the plans previously developed by testing and analyzing the delivered systems. Although, the dividing line between planning and execution is not sharp; the latter begins as soon as possible. For example, the validation of requirements – ensuring they are correct and complete and meet quality standards – is an early lifecycle example of V&V. Another lifecycle consideration is that the operation of the mission, from launch to disposal of the Flight System, is also considered in the V&V planning, by basing validation activities and training scope on the set of mission scenarios that Europa Clipper is expected to execute.

All space missions' V&V campaigns come with specific challenges that arise when there is some aspect of the mission performance that is unique and/or critical to mission success but for which flight-like test configurations are difficult to achieve. Such cases introduce risk, for example by being exceptions to the TAYF principle. On Europa Clipper the key V&V challenges include:

- V&V of the REASON instrument performance, as it will not be possible to test this in a full flight configuration, e.g., using far-field measurements of the instrument on two fully-extended solar arrays in a flight-like environment. REASON performance V&V will instead be based on a series of models of radar performance, supported by tests of non-flight instrument and spacecraft hardware.

- V&V of the suite of tests and models that together provide estimates of the Jovian environments that the Flight System will experience. In particular this includes the static and time-varying magnetic fields at the spacecraft, and the radiation environment in the Europa Clipper science orbit.

- V&V of the Europa Clipper approach to planetary protection, which is framed in terms of the probability of contaminating an icy moon through Flight System impact. Verifying this requirement requires a set of models, with parameters inputs provided by lower-level requirements, e.g., maximum spore counts. These models and inputs will also require validation in their own right.

7. FUTURE WORK

The project is preparing for the PDR follow-on review in January 2019, which will focus on the integrated solar array / REASON preliminary design. After completion of this review, NASA will consider approving the mission for entry into Phase C, the Detailed Design phase.

In the meantime, the season of Critical Design Reviews has begun with an early Propulsion Subsystem CDR in July 2018 (in order to accommodate the long lead time of its propulsion tanks). The bulk of the CDRs will begin in January 2019 and culminating in the Project and Flight System CDR in November 2019.

Finally, detailed planning for Integration and Test, and Verification and Validation is underway. Key reviews of plans for these activities will take place leading up to Project CDR.

8. CONCLUSIONS

Europa, the fourth largest moon of Jupiter, is believed to be one of the best places in the solar system to look for extant life beyond Earth. Exploring Europa to investigate its habitability is the goal of the Europa Clipper Mission. This exploration is intimately tied to understanding the three "ingredients" for life: water, chemistry, and energy. The joint JPL and APL Project team together with the nine science investigation teams have completed a preliminary design and are heading into detailed design and implementation, on an extremely capable flight system which promises to revolutionize our understanding of this enigmatic and tantalizing world.

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BIOGRAPHY

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